Jupiter Solar Mission Alternative Feasibility Study

August 15, 1989

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INTRODUCTION

This study explored the feasibility of replacing the Galileo spacecraft's plutonium-238 fueled radioisotope thermoelectric generators (RTGs) with solar arrays. Technical feasibility considerations were restricted to solar retrofit options that would not substantially compromise Galileo orbiter/probe objectives, configuration or schedule as defined in the Final Environmental Impact Statement for the Galileo Mission (Tier 2) (Ref 1). Thus, other than the power subsystem, the solar retrofit options considered in this study do not necessitate major redesign of the science subsystems, the engineering subsystems including the propulsion module, nor the spacecraft's dual-spin design.

Solar retrofit options for Galileo, however, were not the only considerations of this study. The study also examined some of the technology, cost, and schedule implications of cancelling Galileo and starting over with a completely redesigned spacecraft. This examination occurred in two stages. First, power source technologies were identified and assessed in terms of their applicability to a Jupiter mission. Then, the associated spacecraft design process and its cost and schedule implications were delineated.

GALILEO RETROFIT OPTIONS AND FINDINGS

A variety of considerations encompassing solar intensities, Jovian radiation effects, Space Shuttle and Inertial Upper Stage (IUS) payload capabilities, and, of course, the availability of appropriate solar technology led to focusing the study on a three-panel array option involving two types of solar technology: flight-tested rigid array technology and experimental Advanced Photovoltaic Solar Array (APSA) technology. Array size, mass, and power estimates were then generated for these options. With this information, the impacts of each type of array on the mission design, spacecraft configuration, power subsystem, attitude control subsystem, propulsion subsystem, and science capabilities were investigated.

This investigation revealed that both array options for the originally planned orbiter/probe combination would result in direct and indirect mass increases that would exceed Shuttle/IUS launch capabilities. The study team then investigated the possibility of changing the Galileo mission to a flyby/probe combination. The flyby/probe combination proved inadequate as a means for achieving the mission's science objectives and, for the rigid array option, still entailed a mass in excess of Shuttle/IUS launch capabilities.

The study also examined the schedule implications of the two solar array options. If only the array were considered (exclusive of the resulting system problems), a suitable rigid array would take at least three years to design and manufacture -- too late for either the 1989 or 1991 launch windows. The APSA array would take at least four years to design and manufacture -- again, too late for the 1989 or 1991 launch windows. Beyond the design and manufacturing time, test and spacecraft integration time would also be required. But more critical,

EXECUTIVE SUMMARY

the spacecraft system and subsystem problems arising from either of the solar array options are so severe that, even if the arrays were already available, these problems could not be resolved in time for a 1989 or 1991 launch - or, with reasonable mission risk, not even for several years thereafter.

In view of the insurmountable mass and schedule difficulties associated with a solar retrofit of Galileo, the study team concluded that the only alternative to an RTG-powered Galileo mission would be to cancel the Galileo mission and design a completely new, solar-powered spacecraft for the late 1990's.

EXISTING POWER TECHNOLOGIES AND THEIR IMPLICATIONS FOR THE FUTURE

To the extent that cancellation of the Galileo mission and initiation of a completely new, solar-powered mission to Jupiter might be considered, the study examined the current status of power technologies to see which technologies might best support such a mission. This examination revealed that, among the many solar technologies, Advanced Photovoltaic Solar Arrays (APSAs) show the most promise for a late 1990's Jupiter mission. However, no solar technology demonstrated any viability for missions more distant than Jupiter. As for batteries and fuel cells, no technology exists that will deliver enough energy per pound of mass to make it suitable as a sole power source for any planetary mission. However, a new type of radioisotope thermoelectric converter known as an Alkali Metal Thermoelectric Converter (AMTEC) is being developed that may require far less plutonium-238 fuel than an RTG using thermocouples -- thereby reducing the risk as well as the cost. By the late 1990's, AMTECs might be capable of serving as power sources for outer planet missions.

THE SPACECRAFT DESIGN PROCESS AND ITS NEW MISSION IMPLICATIONS

The spacecraft design process would occur in two stages. The first stage, pre-project development, would involve science objective development, mission design, spacecraft design, and technology development. These pre-project activities would take three to five years. The second stage, normal project development, would involve further spacecraft design, development, integration, and testing. These activities would require an additional four to five years.

For a completely new, solar-powered mission to Jupiter, this process and its associated time requirements indicate that at least seven to ten years of pre-launch activity would be required. Hence, a new Jupiter mission could not be launched earlier than 1996 and probably could not be launched until 1999 or later. On the basis of past project experience, the study team estimated that pre-project development would probably cost $50 million to $100 million. Normal project development would probably cost at least another $1 billion.

CHAPTER I

INTRODUCTION

JUPITER SOLAR MISSION ALTERNATIVE

ENVIRONMENTAL IMPACT STATEMENT

(TIER 2)

- "COMPLETION OF PREPARATION AND OPERATION OF THE GLL MISSION"

- ALTERNATIVES
  - FLY GLL AS IS
  - NO ACTION

- TIME FRAME
  - 1989 - 1991

- ANY SIGNIFICANT REDESIGN IS EQUIVALENT TO "NO ACTION", AS 1991 LAUNCH OPPORTUNITY CANNOT BE MET

INTRODUCTION

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CHAPTER II

GALILEO SOLAR RETROFIT

JUPITER SOLAR MISSION ALTERNATIVE

DESIGN REQUIREMENTS & CONSTRAINTS

- DO GLL SCIENCE
  - DESIRED: PROBE & ORBITER
  - FALLBACK: PROBE & FLYBY

- MINIMUM MODIFICATION TO GLL S/C
  - RETAIN SCIENCE SUBSYSTEMS
  - MINOR CHANGES TO ENGINEERING SUBSYSTEMS
  - RETAIN DUAL-SPIN DESIGN
  - NO PROPULSION MODULE CHANGES

- REPLACE RTGs WITH SOLAR ARRAYS

- RETAIN RHUs

- SHUTTLE/IUS LAUNCH VEHICLE

- 1989 OR 1991 LAUNCH PERIOD

- IDENTIFY SHOW-STOPPERS
  - TECHNICAL
  - SCHEDULE

CHAPTER II

A. MISSION DESIGN

SECTION A

MISSION DESIGN

1. INTRODUCTION

This section reports the results of a mission performance analysis for several Galileo mission options that involve replacing the Radioisotope Thermoelectric Generators (RTGs) with solar arrays. In general, optimistic assumptions have been used in this analysis for calculating launch vehicle and spacecraft performance margins.

2. ASSUMPTIONS AND GROUND RULES

a. **Mission.** For all options, a 1991 launch using the Shuttle/Inertial Upper Stage (IUS) launch vehicle has been assumed. A 0.72 astronomical unit (AU) Venus-Earth-Earth Gravity Assist (VEEGA) trajectory is used for the interplanetary transfer. The "0.72 AU" denotes the perihelion distance after the Venus flyby. Other mission assumptions are as follows:

1) There is no propellant allocation for a velocity change (delta-v) for asteroid flybys.

2) No spacecraft turns are allowed for propulsive maneuvers.

3) No science turns are performed during the satellite tour.

The constraint on spacecraft turns for propulsive maneuvers eliminates the need for batteries to power the spacecraft during the long periods of time spent in an off-sun attitude for maneuvers.

b. **IUS Injection Margin.** For IUS injection margin calculations, the following assumptions have been made:

1) The launch energy requirement is determined in order to provide an 18-day launch period.

2) There is only one revolution about the Earth by the Shuttle before IUS deployment.

3) No provision for a finite launch window is made.

The IUS injected mass capability is taken from Johannesen, J.R., (Ref 2).

c. **Spacecraft Propellant Margin.** Spacecraft propellant margin ground rules are consistent with those in Ref. 2, with the following exceptions:

1) The orbiter dry mass is increased as given below in the section "Orbiter Dry Mass". (See also Stoller, R.L., Ref 3.)

2)   The interplanetary and orbital attitude control propellant allocations are increased as given below in the section "Attitude Control Propellant". (See also Bernard, D.E., Ref. 4.)

3)   A vector-mode penalty of 33% has been assumed for navigation and deterministic delta-v's.

4)   No propellant has been allocated for science turns during the satellite tour.

5)   A Jupiter Orbit Insertion (JOI) penalty of 600 m/s has been included to account for the 8.5 hour delay in the JOI start time due to the additional time required to spin up to 10 rpm.

Propellant margin is computed with respect to 10 satellite encounters at 90% probability. With regard to the spacecraft propulsion system, it has been assumed that there is no change to the propellant capacity of the Retro Propulsion Module (RPM). The maximum usable propellant is therefore 925 kg.

       d.   <u>Orbiter Dry Mass.</u> For the Rigid Array, the dry mass increase is 833 kg. For the Advanced Photovoltaic Solar Array (APSA), the dry mass increase is 298 kg. The injected spacecraft mass for the 1989 baseline mission is 2712 kg.

       e.   <u>Attitude Control Propellant.</u> Updated attitude control propellant requirements are summarized in Table 1. The values for the baseline 1989 mission are also included in this table.

Table I. Attitude Control Propellant Requirements

	Baseline	Rigid Array	APSA
Interplanetary Cruise	50 kg	450 kg	150 kg
Orbital Phase	38 kg	150 kg	50 kg

Table II. Mission Performance Summary

	BASELINE (RTGs)	OPTION 1	OPTION 2	OPTION 3
MISSION OBJECTIVE	ORBITER + PROBE	ORBITER + PROBE	FLYBY + PROBE	FLYBY + PROBE
SOLAR ARRAY OPTION	N/A	RIGID	RIGID	APSA
LAUNCH YEAR	1989	1991	1991	1991
LAUNCH VEHICLE	STS/IUS	STS/IUS	STS/IUS	STS/IUS
TRAJECTORY TYPE	VEEGA	VEEGA	VEEGA	VEEGA
LAUNCH PERIOD, DAYS	40	18	18	18
INJECTED MASS, KG	2712	3550	3220	3010
IUS INJECTION MARGIN, KG	0*	-500	-170	30
S/C PROPELLANT MARGIN, KG	0	-1020**	0	640
JUPITER SATELLITE ENCOUNTERS	10	0	N/A	N/A

* IUS INJECTION MARGIN USED TO PROVIDE LAUNCH WINDOW AND DEPLOY REVS

** PROPELLANT EXHAUSTED DURING JOI

☐ = SHOWSTOPPER OR SEVERE DEFICIENCY

3. RESULTS

Three options were investigated. Option 1 is an orbiter plus probe mission with the Rigid Array. Option 2 is a flyby plus probe mission (no satellite tour) with the Rigid Array. Option 3 is a flyby plus probe mission with the APSA. The results are presented in Table 2. In addition to the three solar array options, the table includes data for the 1989 baseline mission as well.

Option 1 is not feasible, because it has extremely large negative values for both IUS injection margin and spacecraft propellant margin. The negative IUS margin is caused by the additional dry mass of the solar array and supporting hardware. The negative propellant margin is caused primarily by the large increase in attitude control propellant (see Table 1) and the delta-v penalty for delaying the start of JOI. For this option, spacecraft propellant is exhausted before completion of JOI. It should be pointed out, however, that even if the JOI delta-v penalty were not included, propellant still would have been exhausted before completion of JOI.

Option 2 is not feasible because it has a negative IUS injection margin. Propellant margin for this option is zero. In order to minimize the IUS shortfall, approximately 330 kg of propellant has been off-loaded from the RPM. It is not possible to eliminate the IUS shortfall by off-loading even more propellant, because propellant margin would then become negative. In any case, this option would probably be considered unacceptable from a science point of view because the Jupiter satellite tour has been eliminated.

Option 3 has a small positive IUS injection margin and a very large positive propellant margin. The IUS margin can easily be increased to provide for multiple deploy revs and a finite launch window by off-loading propellant. Off-loading one kilogram of propellant will increase IUS injection margin by one kilogram and decrease propellant margin by slightly less than one kilogram. The propellant margin decrease caused by off-loading propellant would not be a problem, because there is an excess of propellant margin for a fully loaded RPM. As was the case for Option 2, however, Option 3 would also probably be considered unacceptable, because it requires giving up the Jupiter satellite tour.

4. SUMMARY AND CONCLUSIONS

Three Galileo mission options that involve replacing the RTG's with solar arrays and delaying launch to 1991 have been evaluated for mission performance. Option 1, an orbiter plus probe mission with Rigid Array, would not be feasible because the IUS injection margin and spacecraft propellant margin are both negative. Option 2, a flyby plus probe mission with the Rigid Array, would also not be feasible because of negative IUS injection margin. Option 3, a flyby plus probe mission with the APSA, would have positive margins, but giving up the satellite tour would make this option unacceptable from a science point of view.

CHAPTER II

B. SPACECRAFT CONFIGURATION

GALILEO SOLAR RETROFIT TASK
SPACECRAFT CONFIGURATION

CONFIG. & STRUC. CHANGES TO REPLACE RTG'S WITH SOLAR PANELS

- 3 ARRAYS REQ'D FOR INERTIAL BALANCE OF ROTOR

- EACH ARRAY MUST ARTICULATE ABOUT ITS LONG AXIS FOR TEMP. CONTROL; AND ALSO UP & DOWN FOR WOBBLE CONTROL

- BEEF UP BUS TO SUPPORT ARRAYS IN LAUNCH CONFIGURATION

- STOWED ARRAYS CAN BE LATCHED ONLY OVER THE BUS FOR LAUNCH, WHICH WILL MOST LIKELY REQUIRE REDESIGN OF ALL OF THE CORE STRUCTURE (IE. BUS, RPM, DESPUN SECTION, ADAPTER)

- AT LEAST 3 BAYS OF ADDITIONAL ELECTRONICS PACKAGING VOLUME MUST BE ADDED TO THE SPACECRAFT

SECTION B

SPACECRAFT CONFIGURATION

1.   INTRODUCTION

Major configuration and structural changes would be required to replace Radioisotope Thermoelectric Generators (RTGs) with solar panels. In order to provide a rotationally symmetric rotor, three solar arrays would be required. Having only one or two arrays would provide unacceptable inertial properties. Each array would need to be capable of articulation about its long axis in order to reduce the solar flux on the panel when close to the sun. This would be required for both temperature control reasons and power control reasons. The angle of off-sun pointing will vary with distance from the sun. Each array would also have a linear boom actuator to tweak its position in the $\pm$ Z direction. This is an attitude control requirement for wobble control.

In order to support the three massive solar arrays for launch, the eight-bay electronics bus would have to be augmented structurally to sustain higher loads. Since it appears that the stowed arrays can only fit above the bus for launch (positioned around the stowed High Gain Antenna (HGA)), it is most likely that all of the core structure would have to be augmented. This would probably require redesign of not only the bus, but also the Retro Propulsion Module (RPM) structure, the despun section, and the adapter. It is possible that the Inertial Upper Stage (IUS) structure would also need to be augmented for higher loads.

GALILEO SOLAR RETROFIT TASK
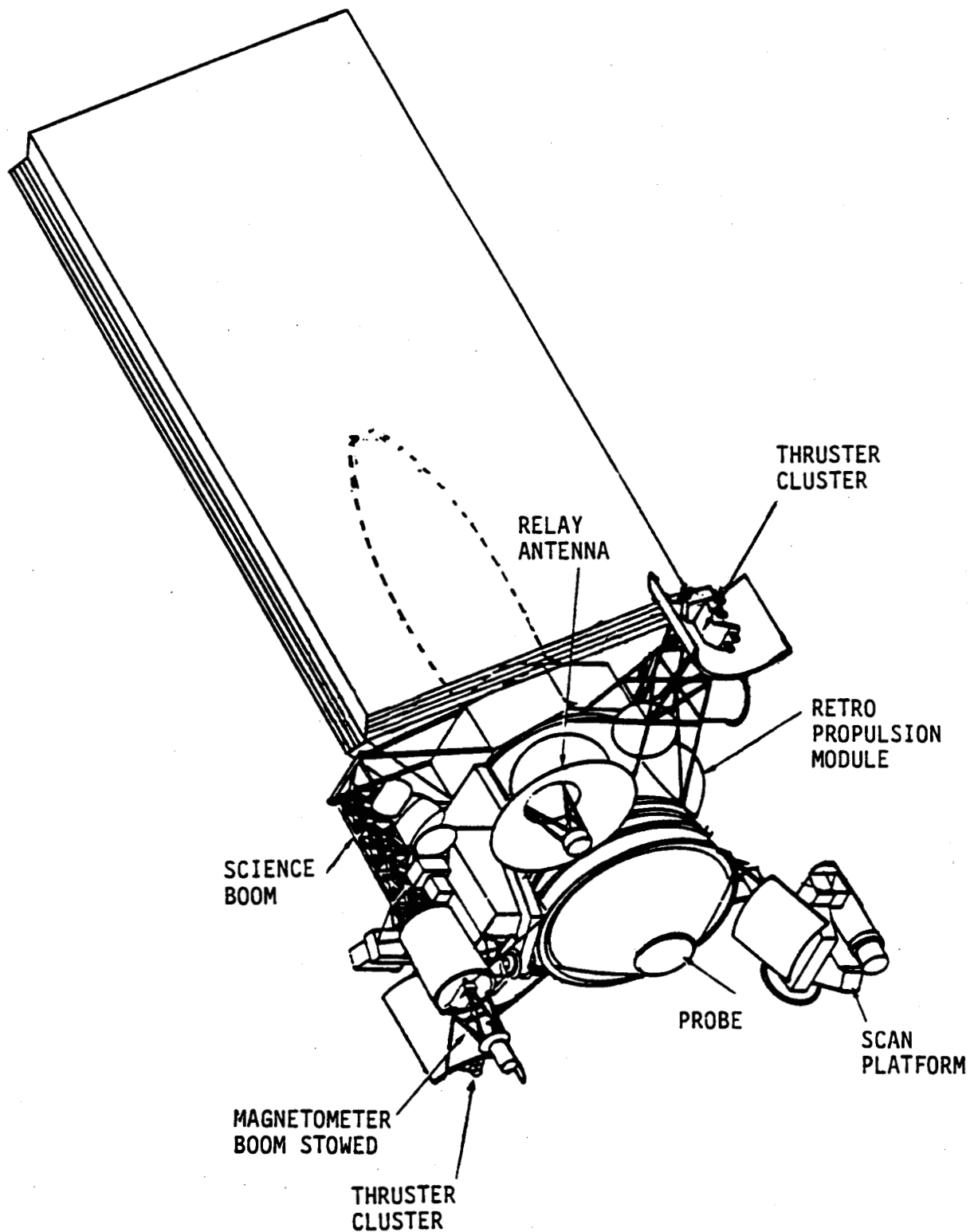## SPACECRAFT CONFIGURATION

Figure 1. Galileo Spacecraft with Rigid Solar Arrays in Stowed
(Launch) Configuration
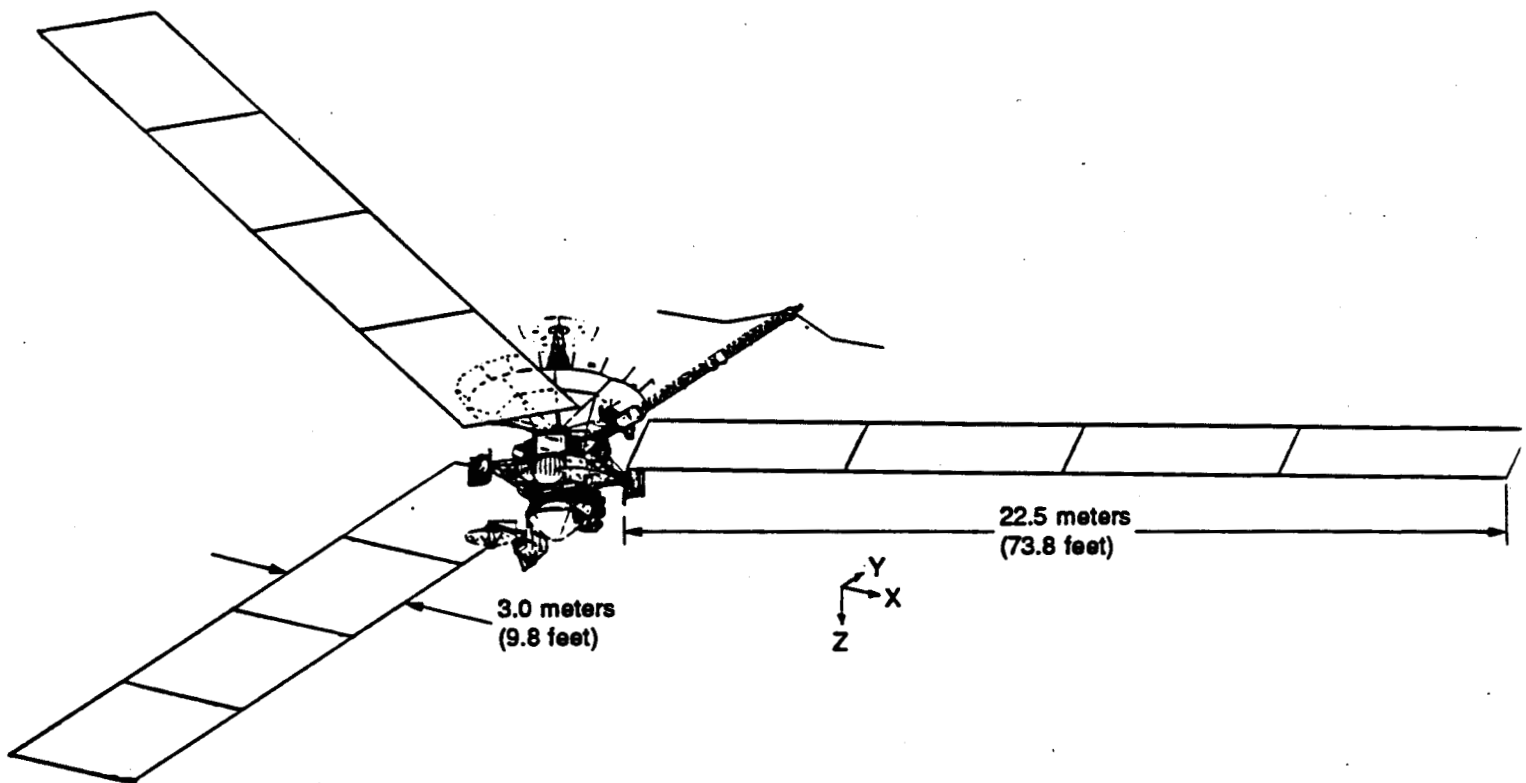
GALILEO SOLAR RETROFIT TASK
SPACECRAFT CONFIGURATION

Figure 2.   Galileo Spacecraft with Rigid Solar Arrays Deployed

GALILEO SOLAR RETROFIT TASK
SPACECRAFT CONFIGURATION

CONFIGURATION & STRUCTURAL ISSUES

- TECHNICAL SHOW STOPPERS
    - NONE YET IDENTIFIED OTHER THAN MASS CONSTRAINTS
- SCHEDULE SHOW STOPPERS
    - MODIFICATION OF S/C CORE STRUCTURE FOR HIGHER LOADS
    - NEW IUS LOADS & DYNAMICS
    - NEW SHUNT RADIATOR SYSTEM
    - DESIGN & QUAL. OF RIGID SOLAR PANEL DEPLOYMT. MECHANISMS
    - PROCUREMENT OF RIGID SOLAR PANEL ASSEMBLIES
- OTHER MAJOR CONCERNS
    - 1.5 g AVERAGE LOAD ON SOLAR ARRAYS AT 10 RPM
    - PLUME IMPINGEMENT FOR +P1A & +L1B THRUSTERS
    - PACKAGING OF BATTERIES AND ADDITIONAL ELECTRONICS
    - SOLAR PANELS IN TEMP. CONTROL F.O.V. FOR BUS
    - SCI. INSTR. AND RELAY ANTENNA F.O.V. BLOCKAGE BY SOLAR ARRAYS
    - BUS SUNSHADE MUST BE DEPLOYABLE TO CLEAR STOWED ARRAYS

2. CONFIGURATION AND STRUCTURAL ISSUES

a.   <u>Technical Show Stoppers.</u> In the limited course of this study, no technical show stoppers were found other than mass constraints on the structural design.

b.   <u>Schedule Show Stoppers.</u> Modification of the spacecraft core structure could not take place within the framework of meeting a 1991 launch. Other schedule show stoppers which have been identified are qualifying the IUS for higher loads and dynamics, installing a new shunt radiator system, design and qualification of the deployment mechanisms for new rigid solar arrays, and design, procurement, installation, and testing of the rigid solar panel assemblies.

c.   <u>Other Major Concerns.</u> A number of issues are considered to be major concerns although they cannot be shown to be show stoppers at this time. At 10 RPM, the solar arrays will see an average load of 1.5 g. Rigid solar arrays can be designed to sustain this load at the expense of a mass penalty; however, the implications of such a requirement on a SAFE or APSA design are unknown. There would be a plume impingement problem with solar arrays for the +P1A and +L1B thrusters on the RPM. It has not been determined what kind of thruster layout redesign would be required to resolve this. Packaging of batteries and additional electronics would require providing several new bays on the spacecraft. It would be difficult to find a location for them on the rotor, and placement on the despun section would pose the problem of power transmission across the spin bearing. The solar arrays would be in the thermal field of view of radiating surfaces on the electronics bus. It is not known how serious this might be for temperature control of the electronics. Additionally, the arrays would present field of view blockage for the science instruments and the probe relay antenna. In order to provide space for storage of the solar arrays during launch, the bus sunshade would have to be redesigned to be deployable.

CHAPTER II

C. SPACECRAFT POWER

GLL SOLAR ARRAY RETROFIT - PPS

JPL
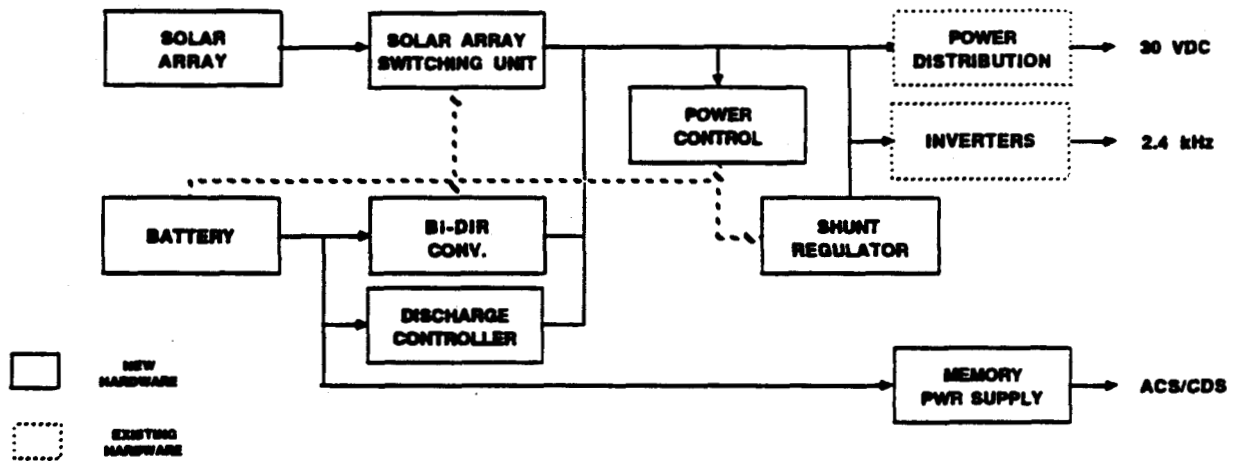
POWER ASSY.	MASS INCREASES (kg)	COMMENTS
ELECTRONICS	25	GLL WAS 43kg, NEW SYS IS 68kg
BATTERY	140	SUPPORT 4 Hrs. @ 450W. Ni-Cd SELECTED DUE TO VOLUME
SOLAR ARRAY	550	RTG MASS WAS 110kg, S/A IS 650kg
TOTAL	715	

SECTION C

SPACECRAFT POWER

1. **GLL SOLAR ARRAY POWER/PYRO SUBSYSTEM (PPS) REQUIREMENTS AND ASSUMPTIONS**

The PPS is required to process and deliver an average of 500W of power to the Galileo Spacecraft (GLL S/C) loads during the Jupiter tour. The existing power distribution design requires a shunt regulated 30 Vdc and a 2.4 kHz square wave to be distributed to the user loads. It was assumed that the occultation time at first encounter ($\approx$4 hrs. @ 450W) would be the driver for sizing the batteries. There are other occultations in current planning that extended this time to the 7-14 hr. range. Minimum battery recharge power was considered to be shared between the solar array and the existing power profile. Solar array output of 650W less 150W of recharge power would provide about 500W to S/C loads during battery recharge.

2. **PPS OVERVIEW**

These requirements and optimistic assumptions will impose a new PPS design for GLL retrofit. As shown in the Block Diagram only the inverters and power distribution could be retained. Tight bus regulation specific to user loads requires a shunt regulator design. The impact of large solar array power during the inner planet Venus-Earth-Earth Gravity Assist (VEEGA) tour coupled with a large array voltage range would force a highly responsive Solar Array Switching Unit (SASU) to be coupled to a Shunt Regulator (SR) that has increased power dissipation capability. The SASU would operate upon SR control to maintain total array capability at a power level above the total bus load demand by shorting out excess array capacity. Array voltage increases (outbound trajectory) would be handled by the SASU via a series to parallel switch of array segments between Mars and Jupiter. An added SR stage would be required to allow for loop response in the interaction required to maintain a 30 Vdc bus. Planet occultation time at Jupiter would require battery energy storage. A three of four battery redundancy scheme was used for gross sizing. The batteries would be at a voltage above the 30 Vdc bus so that the existing transient response requirements could be maintained. Battery power would be delivered to the bus and recharge power supplied from the bus via 4 Bidirectional Converters. GLL computer memory keep-alive power would then be supplied from the battery and conditioned in the Memory Power Supply. Power Control would house the interfaces to the power bus from the arrays and the batteries. A new fault protection architecture for bus and source faults would have to be designed and implemented. Fault protection, control and sensing of three solar arrays and four batteries would ripple through the entire S/C autonomous control scheme.

GLL SOLAR ARRAY RETROFIT - PPS

JPL

POWER ELECTRONICS ISSUES

o SHOW STOPPERS
- PPS MASS INCREASE IS 165kg
  - BATTERY - 140kg, 68L (Ni-Cd @ 50% DOD, 450W-4 Hrs, 3 OF 4 BATTERY)
  - PWR ELECTRONICS - 25kg,1/3 BAY
- SAF DELIVERY TIME MINIMUM IS 2.5 YEARS

o PROBLEMS
- INBOUND OFF SUN OPS vs S/A GRANULARITY, SASU & SHUNT REG. RESPONSE
- S/A & BATTERY CAPABILITY IMPACT SAFETY & INVERTER FAULT PHILOSOPHY
- FAULT PROTECTION MODIFIED FOR OVER-POWER & BATTERY
- BATTERY SIZE vs OTHER OFF-SUN TURNS
- GLL PWR BUS SPEC. TO USERS PREVENTS OPTIMUM S/A & PPS
- SOLAR ARRAY SWITCHING REQ. (SERIES TO PARALLEL)
- INCREASED CMD/TLM REQUIRED OF CDS
- BATTERY CONV. NOT COMPATIBLE WITH SCI. (PWS)

3.   MASS, VOLUME AND SCHEDULE IMPACTS

The mass calculations are based upon estimates for PPS assemblies and Ni-Cd batteries required to support the requirements and assumptions outlined above. Several PPS components are Mariner Mark II (MMII) preliminary designs and have been scaled up in power to meet GLL needs. The remaining electronics masses were estimated based upon scaling the existing GLL hardware. Existing PPS electronics mass is 43 kg, and the additions and deletions of electronic components result in a new total estimate of 68 kg. Battery mass represents a total capability and has no allowance for the realities of discrete battery masses.

Volume impact would also be a big issue for the S/C configuration retrofit. Battery volume would be a minimum of 68 liters with no allowance for actual battery sizes or mounting and cables/connectors. The summation of additions and deletions of power electronics hardware would add volume in excess of 1/3 bay. The total estimated volume required would exceed 3 GLL bays.

Optimistic estimates of schedule for PPS assemblies and batteries are 2.5 years to get the flight hardware to S/C assembly. This allows only 6 months to get on contract and complete a new power system design. Hardware fabrication and assembly test time would be 2 years. Current delivery times for flight qualified parts would be in excess of one year and the requirements ripple effect through S/C system to subsystem would realistically take in excess of one year to achieve a workable PPS design.
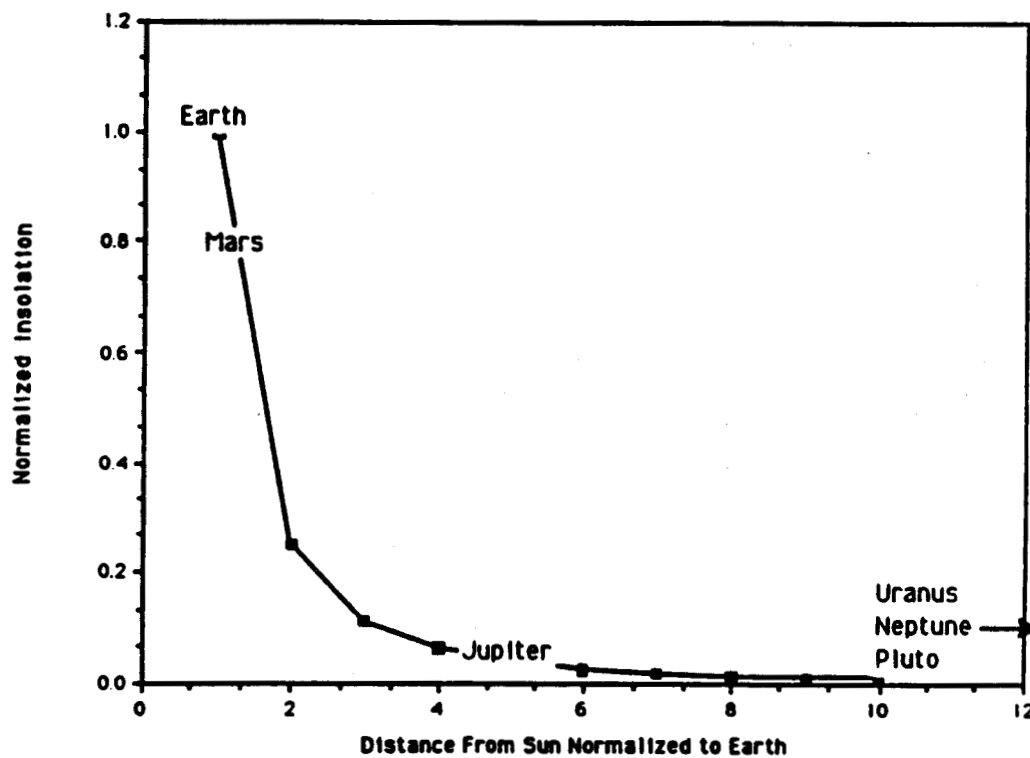
4. PPS PROBLEMS AND ISSUES

a. To maintain reasonable temperatures the array would be pointed off sun during the inbound tour. The change in array power, increasing inbound and then decreasing outbound, would require shorting of a majority of the array capability. The VEEGA inbound tour presents a severe constraint upon the granularity of the solar array sections and the speed at which the SR and SASU must respond to any change in solar array sun angle. SR capacity would be increased to about 750W to accommodate array section switching in discrete quantities.

b. The added source currents available from an array of this size and 4 batteries would impact safety due to fault currents far in excess of the short circuit capability of the Radioisotope Thermoelectric Generators (RTGs). Fault concepts for GLL and wire sizes will have to be redesigned to accommodate the worst case fault capability of the available array/battery short circuit current during fault modes. PPS inverters were designed to operate into a short circuit to clear faults. This concept was viable with a current limited source. Now that there is essentially unlimited current to the inverter, fault philosophy will require a redesign.

c. S/C and PPS autonomous fault protection must be redesigned to accommodate array over-power and battery power conservation. The present S/C fault protection design has been derived from achieving a known power state for any conceivable fault scenario. Retrofit of the RTGs with solar and batteries would present both the PPS and the S/C with the problem of not having control of what state the power source is in after an in-flight anomaly.

d. Battery size has been selected with the first Jupiter occultation as the driver for duration. There may be other such events in the VEEGA tour or the Jovian tour that exceed this duration.

e. The specification to user loads would have 30 Vdc plus a small allowance for long term drift as the upper limit. An optimum design for a solar array PPS would utilize a switching regulator for bus regulation with peak power tracking capability. This approach has failure modes that would put a transient overvoltage on the bus. A new PPS design would specify this characteristic to the user loads such that their designs of load converters would handle this transient.

f. Due to array cooling on the outbound journey, the series array segments would have to be switched to a parallel configuration somewhere between Mars and Jupiter. This switch would optimize the array output to the Jupiter tour within the limits of voltage variations. The implication of this is that the array cannot be optimized for power capability without a peak power tracking PPS.

g. Retrofit of two RTGs to three solar arrays and four batteries would increase the commands to PPS sources by a factor of about three. These added power sources would also increase the telemetry (TLM) measurements by a factor greater than four due to individual battery voltages/temperatures and the array temperatures/currents.

h. The bidirectional converters for battery interface to the 30 Vdc bus may be incompatible with some of the science instruments on the Galileo Plasma Wave Subsystem (PWS). GLL has an electromagnetic interference (EMI) environment designed around the 2.4 kHz inverter. The switching intervals are blanked out of the measuring instruments as frequency is fixed and clocked through the Command and Data Subsystem (CDS) to all user loads.

Solar Array Issues

Insolation As A Function of Distance From Sun

Figure: Normalized Insolation vs. Distance From Sun Normalized to Earth. Data points labeled Earth (~1.0), Mars (~0.8), Jupiter, Uranus, Neptune, Pluto.

5. SOLAR INSOLATION (W/m^2) at Jupiter

The amount of power received by a given area falls off as $1/r^2$, where r is the distance from the sun in AU. Thus, at Earth there is a normalized unit of 1 solar intensity. By the time a spacecraft reaches Jupiter only 4% or 1/25th of the energy at the Earth is available to produce useable power. At even further distances from the Sun for the planets of Saturn and beyond, there is virtually no solar intensity available to produce power. Jupiter is about the extreme of where solar photovoltaics can be used as a viable nonisotopic power source.

Solar Array Issues

Solar Array Sizing

Power Required at Jupiter - 650 Watts

Accounting for LILT and Radiation Design Margin - 1066 Watts

Earth Equivalent Size - $1066 \cdot (5.3)^2$ - 30kW Array

Array Size ~ 210 m^2

6. ARRAY SIZING

The size of the array is determined by the power required at Jupiter. Assuming that an average of about 470-500 W is required in orbit at Jupiter, an additional 150 W would be required to recharge the batteries that will be used to power the spacecraft during off-sun activities. Thus the power required would be 650 W.

To account for the increased efficiency (up to 50%) of solar cells due to the colder temperatures at Jupiter, and to account for any degradation due to low intensity, low temperature (LILT) effects, a 20% net increase in performance over Earth was assumed. This was based upon the data from the NASA TM-78253 (Ref. 5) detailing the range of performance for LILT. In addition, a radiation degradation margin of 2 was assumed. Thus, 1066 W would be the power design point.

Then to account for the solar intensity effect, the design point must be multiplied by 5.3 squared to get the actual array size of 30 kW. Current industry designs when completed achieve 145 W/m^2, leading to an area of about 210 m^2.

Cell Choice

Silicon

Tremendous amount of history

Largely successful

Radiation effects well understood

Low intensity, low temperature effects categorized

GaAs

Little flight experience

Limited manufacturing capability in US

Initial analysis shows limited advantage for Jupiter environment

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Conclusion - *Silicon is cell of choice due to its experience and wealth of operational knowledge. More detailed studies of GaAs option needed to verify Silicon choice.*

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7.   CELL CHOICE

Silicon has a tremendous amount of flight history over the years. Silicon cells are currently flying on Magellan and will fly on Mars Observer and Ocean Topography Experiment (TOPEX). In addition, the radiation effects upon Silicon and LILT effects are well understood. Such data is available in JPL Publication 82-69 (Ref. 6).

GaAs on the other hand has limited flight experience. A recent trade study for Mars Observer and TOPEX determined that GaAs is still too risky for a current mission. Furthermore, by looking at radiation degradation charts in JPL Publication 82-69, Addendum 1, there appears to be no advantage in going to GaAs cells for the Jupiter environment.

Thus, Silicon cells are still the only choice for a solar array, especially for a launch in 1991.

Solar Array Issues

Rigid Array (Current State of Art)

Size	200 sq m
Mass	650 kg
Manufacturability/Schedule	3 years (1 yr design; 2 year build)
Technical Risk	Panels/wings this size not flown before.
Deployment	Difficult
Conductivity	Surface charging is an issue.
Issues	Low intensity, low temperature (LILT) effects add greatly to risk and cost of mission.

Showstoppers - *Solar Array mass*

Solar Array delivery for 1991 launch

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8.   CURRENT PLANAR ARRAY STATE OF THE ART

Due to the power required by the array at Jupiter, the mass would be extremely large for a state-of-the-art array. Current designs have shown a specific mass of about 35 W/kg. This would yield a solar array mass of 650 kg.

Typical build schedules for planar arrays are anywhere from 3 to 4 years. As a point of reference, TOPEX (which launches in 1992) is currently undergoing panel qualification testing 3 years before launch. Thus, the delivery of the necessary array for a 1991 Galileo launch would be extremely unlikely.

Due to the large size of the array, surface charging of the cover glass would be an issue. This surface changing could impact some of the on-board science instruments.

The increased cost for testing at the cell and string level to minimize LILT effects could also be prohibitive. Cells and strings would need to be tested at -150°C to insure proper operation of the array at Jupiter. Such testing has never been done before, other than in a laboratory environment.

In summary, the current technology would be too heavy for a retrofit application and could not be delivered for a 1991 launch.

Solar Array Flight Experiment Flexible Array (SAFE)
(Current Advanced Technology)

Size	**210 sq m**
Mass	**420 kg**
Manufacturability/Schedule	**4 years (1 yr design; 3 year build)**
Technical Risk	**SAFE flown as partial array on shuttle. Only flown on Lockheed "Black" programs.**
Deployment	**Array unfurled from canister**
Conductivity	**Surface charging is an issue**
Issues	**Low intensity, low temperature (LILT) effects add greatly to risk and cost of mission**

Showstoppers - *Solar Array mass*

Solar Array delivery for 1991 launch

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9. ADVANCED FLEXIBLE ARRAY TECHNOLOGY

The SAFE array has flown as a shuttle experiment. Its design was proven to have a specific power of 60-70 W/kg. Thus, the mass of this array would be decreased by a factor of 2.

However, due to the new nature of the array, its lack of a design for 30 kW, and its unknown performance for planetary missions, it also would be an unlikely candidate for a 1991 launch.

Due to the large size of the array, surface charging of the cover glass is an issue. This surface changing could impact some of the on-board science instruments. The use of interconnected conductive covers to ameliorate this charging problem has not been demonstrated for flexible fold-up arrays.

The increased cost for testing at the cell and string level to minimize LILT effects could also be prohibitive. Cells and strings would need to be tested at -150°C to insure proper operation of the array at Jupiter. Such testing has never been done before.

In summary, even the advanced technology would be too heavy for a retrofit application and could not be delivered for a 1991 launch.

Solar Array Issues

Advanced Photovoltaic Solar Array (APSA)
(Advanced Technology)

Size	210 sq m
Mass	210 kg
Manufacturability/Schedule	1 yr design; 3 year build
Technical Risk	Now in ground demonstration. Not ready for flight till mid 90's launch.
Deployment	No retractable design
Conductivity	Surface is an issue
Issues	Low intensity, low temperature (LILT) effects add greatly to risk and cost of mission. For Galileo, boom must be stiffened.

Showstoppers - *Solar Array delivery for 1991 launch*

10. FUTURE FLEXIBLE ARRAY TECHNOLOGY

The APSA program is just now in the ground demonstration phase. A prototype is being built that shows a specific power of 130 W/kg. This decreases the solar array mass by another factor of 2 to the lowest mass array.

However, due to its uncertainty of design, and its lack of technology readiness until the early to mid 1990's, this candidate is also not acceptable for a 1991 launch.

Due to the large size of the array, surface charging of the cover glass is an issue. This surface changing could impact some of the on-board science instruments.

The increased cost for testing at the cell and string level to minimize LILT effects could also be prohibitive. Cells and strings would need to be tested at -150°C to insure proper operation of the array at Jupiter. Such testing has never been done before.

In summary, the APSA array could not be delivered for a 1991 launch.

Spacecraft Power Issues

Summary

Electronics Showstoppers

Technical - Fundamental redesign of subsystem and fault
protection approach

Schedule - New electronics can not be delivered for 1991 launch

Solar Array Showstoppers

Technical - Rigid array mass

SAFE array mass

Schedule - Solar array can not be delivered for 1991 launch

CHAPTER II

D. SPACECRAFT ATTITUDE CONTROL

GALILEO SOLAR RETROFIT TASK
ATTITUDE CONTROL AND DYNAMICS

CONTROL AND DYNAMICS ISSUES

TECHNICAL SHOW STOPPER:

INCREASED ATTITUDE CONTROL PROPELLANT REQUIREMENTS

•SOLAR PANEL MASS AND SIZE ⇒LARGE INERTIA AND ANGULAR MOMENTUM

⇒SLOW PRECESSION AND SPINUP

AND

HIGH PROPELLENT CONSUMPTION

•PARAMETERS:	GLL MISSION	RIGID PANEL SOLAR ARRAYS	APSA SOLAR ARRAYS
SPACECRAFT SPIN INERTIA (Kg-m^2)	6000	108000	36000
TIME TO SPIN UP (Hours)	0.5	9.0	3.0
ATTITUDE CONTROL PROPELLANT (Kg)	` 78	600	200

•PROPELLANT REDUCTION OPTIONS CONSIDERED AND ELIMINATED:

—LOWERING SPIN RATE

—CHANGING SOLAR PANEL CONFIGURATION TO ANNULAR ARRAY

—RECONFIGURING THE THRUSTER CLUSTERS FOR LARGER MOMENT ARMS

—PUT PANELS ON (second) DESPUN PLATFORM

—3-AXIS ATTITUDE CONTROL

SECTION D

SPACECRAFT ATTITUDE CONTROL

1.   INTRODUCTION

In order to provide sufficient power for the Galileo spacecraft at Jupiter, three solar panels, each 3 meters by 22.5 meters would need to be added to the rotor. Such large solar panels would give the spacecraft a large spin inertia and therefore a large angular momentum.

2.   ANGULAR MOMENTUM

The angular momentum defines the time and propellant required to turn or spin up the spacecraft. For current technology rigid panel solar arrays, the angular momentum is so large that 600 Kg of attitude control propellant would be needed to complete the mission and 9 hours would be required to change the spacecraft spin rate from 3 to 10.5 revolutions per minute (rpm). When the lighter Advanced Photovoltaic Solar Arrays (APSAs) become available, these numbers may be cut by a factor of three.

3.   ATTITUDE CONTROL PROPELLANT USAGE

This level of attitude control usage would be excessive, despite savings made by eliminating all but four navigation turns. The four turns retained would be test, probe release, Orbit Deflection Maneuver (ODM), and Perijove Raise Maneuver (PJR). All Trajectory Correction Maneuvers (TCMs) and Orbit Trim Maneuvers (OTMs) would be performed in the previously less efficient vector mode. A number of ideas for reduction of propellant usage were considered, but all would require more extensive modification of Galileo, with attendant schedule problems. These ideas are listed below with a brief discussion:

a.   Lowering spin rate: This would cause a propellant unporting problem, possible modification of the star scanner, and extensive modification to the Attitude and Articulation Control Subsystem (AACS) software.

b.   Changing solar panel configuration to an annular array: This would lower the spin inertia by a factor of four, but no deployment scheme for such an array exists at the present time.

c.   Reconfiguring the thruster clusters for larger moment arms: This would require propulsion module changes.

d.   Put the panels on a second despun platform: many open issues, would require a preliminary feasibility study; solution for all-spin mission portions probably would require retraction of the panels.

e.   3-axis attitude control: start from scratch on AACS and propulsion subsystems.

GALILEO SOLAR RETROFIT TASK
ATTITUDE CONTROL AND DYNAMICS

CONTROL AND DYNAMICS ISSUES (CONT.)

SCHEDULE: SEVERE PROBLEM for 1991 launch
- •FLUX/TEMPERATURE CONTROL FOR PANELS
  - —new actuators and sun sensors
  - —new control algorithm development
  - —additional fault protection algorithms
  - —software development/test

OTHER ATTITUDE CONTROL AND DYNAMICS CONCERNS
- •DYNAMIC STABILITY
  - —minimum appendage stiffness
- •STAR TRACKER
  - —stray light FOV
- •WOBBLE CONTROL
  - —articulate solar panels for dynamic balance
- •ATTITUDE AND MANEUVER CONTROL
  - —algorithm and S/W changes
- •PLATFORM POINTING CONTROL
  - —algorithm and S/W changes
  - —pointing accuracy degredation
- •SOLAR TORQUES

4. ARTICULATION CONTROL SYSTEM REDESIGN

The second most severe attitude control and dynamics problem would be the need to develop an entire new solar panel articulation control system in time for a 1991 launch. The panels would need to be articulated about their long axis for thermal control and about their attach points for wobble control. New actuators, sensors, and control electronics would have to be designed and procured, and an entire cycle of algorithm definition, software development, subsystem testing, and system integration would need to be completed. This is considered a severe schedule problem, but not a show stopper.

5. ADDITIONAL ATTITUDE CONTROL AND DYNAMICS ISSUES

Other attitude control and dynamics issues include:

a. Dynamic stability: all panel flexible modes must be kept above 0.3 Hz for stability at 10 rpm. Additional structural mass may be required to provide this.

b. Star Tracker field of view (FOV): no part of any solar panel may intrude into the star tracker's conical stray-light FOV (15° half angle.)

c. Wobble Control: the solar panels must be articulated for dynamic balance. Modification of the existing algorithms and software would be needed.

d. Attitude and Maneuver control: the large change in inertia would force algorithm and software changes.

e. Platform pointing control: algorithm and software changes would be needed to handle the new flexible spacecraft. In spite of these changes, pointing accuracy degradation is expected.

f. Solar Torques: attention would need to be paid to balancing the solar torques, especially during the inner solar system portion of the flight.

GALILEO SOLAR RETROFIT TASK
ATTITUDE CONTROL AND DYNAMICS

AACS PROPELLANT REQUIREMENTS (KG)

	GLL MISSION	RIGID PANEL SOLAR ARRAYS (18× GLL INERTIA)	APSA SOLAR ARRAYS (6× GLL INERTIA)
CRUISE			
SUN/EARTH POINTING OF HGA	15	280	93
SPIN UP/DOWN (Test/Probe Rel./ODM/Maint.)	9	155	52
NAV TURNS (Test/Probe Release/ODM only)	1	17	6
OTHER (deleted from solar retrofit mission)	21	0	0
CRUISE SUBTOTAL (KG)	**46**	**450**	**150**
JUPITER ORBIT			
SUN/EARTH POINTING OF HGA	2	34	11
SPIN UP/DOWN (JOI/PJR/Maint.)	6	110	36
NAV TURNS (PJR)	0.5	9	3
OTHER (deleted from solar retrofit mission)	24	0	0
ORBIT SUBTOTAL (KG)	**32**	**150**	**50**
MISSION TOTAL (KG)	**78**	**600**	**200**

6. PROPELLANT REQUIREMENTS

The AACS propellant requirements are broken down into mission phase and maneuver type. Three quarters of the propellant would be needed before Jupiter Orbit Insertion (JOI) and one quarter afterwards. Approximately half of the total would be needed to keep the High Gain Antenna (HGA)/sunshade pointed at the sun or the Earth as the spacecraft and planets move in their orbits. The other half would be used for spin rate changes to allow probe release, ODM, JOI, and PJR to be performed at 10.5 rpm. Less than 5% would be allocated for navigational turns.

CHAPTER II

E. SPACECRAFT PROPULSION

PROPULSION MODULE - ASSUMPTIONS

- ASSUME UNCHANGED PROPELLANT CAPACITY:

  - o LV CAPABILITY WON'T ALLOW INCREASED MASS FOR SOLAR POWER AND ADDITIONAL PROPELLANT

  - o ENLARGING INTEGRATED TANK/STRUCTURE REQUIRES TOTAL REDESIGN. NOT POSSIBLE FOR 1991

  - o GLL 10-N THRUSTERS LIMITED TO CURRENT LIFE-/THROUGHPUT

  - o ANY MAJOR CHANGE REQUIRES NEGOTIATIONS AND CONCURRENCE WITH THE FRG SUPPLIER OF PROP MODULE

SECTION E

SPACECRAFT PROPULSION

1.   CURRENT CONDITIONS AND STATUS

The Galileo propulsion subsystem is a highly integrated design with tanks, supporting structure, spin bearing assembly and thruster assemblies having common elements.

2.   RETROFIT ASSUMPTIONS AND CONSTRAINTS

A primary assumption for the 1991 mission solar retrofit is that the propulsion propellant capacity would not be increased to accommodate increased dry mass of the solar powered design. The reasons for this assumption are:

a.   The 1989 Launch Vehicle capability would not permit both the mass increase for the solar design and added propellant mass.

b.   Technical rationale also supports this decision. Enlarging the integrated tank/structure assembly would require a total propulsion redesign; which would not be achievable in the 1991 schedule.

c.   Finally, any schedule, scope, or technical changes would be difficult with this subsystem, which is supplied by the German Government. The German Government may not be receptive to participating in a redesign.

JPL

PROPULSION MODULE - ISSUES

● CRITICAL ISSUES REQUIRE DETAILED ANALYSIS

1. PLUME IMPINGEMENT ON SOLAR PANELS

   - SURFACE CONTAMINATION

   - PERFORMANCE LOSS, MANEUVER ERRORS

   - UPSETTING TORQUES

2. EXTENDED MANEUVER SEQUENCES

   - 10-N FIRING CONSTRAINTS LIMIT SEQUENCES

   - RETROFIT S/C, HIGH MASS/INERTIAS, MAKE
     SEQUENCE TIMES LONGER

   - TIME-CRITICAL SEQUENCES (JOI) REQUIRE
     TOTAL REDESIGN

3.   CRITICAL ISSUES

        Critical issues associated with using this propulsion module for the
solar retrofit mission are:

        a.   Plume Impingement on Solar Panels.   The lateral (L1B) and
              precession (P1A) thrusters would be used in a large fraction
              of the spacecraft maneuvers. These thrusters would be canted
              towards the solar panels, and the plume impingement on the
              panels would cause the following:

              1)   Dynamic stability problems

              2)   Propulsive performance losses

              3)   Contamination of panel surfaces

        b.   Excessive Thruster Firing Times. The 10 newton (N) thrusters
have constrained operating times due to thermal problems. These problems make
current GLL sequences difficult and time-consuming. The large mass and inertia
of the solar design would increase maneuver times significantly; for example,
a spinup from 3 to 10 revolutions per minute (rpm) would take 18 times longer
due to the higher spin inertia caused by the panel mass. These times would make
some critical sequences impractical, and would require large increases in
propellant mass.

        These issues cannot be resolved without detailed spacecraft design and
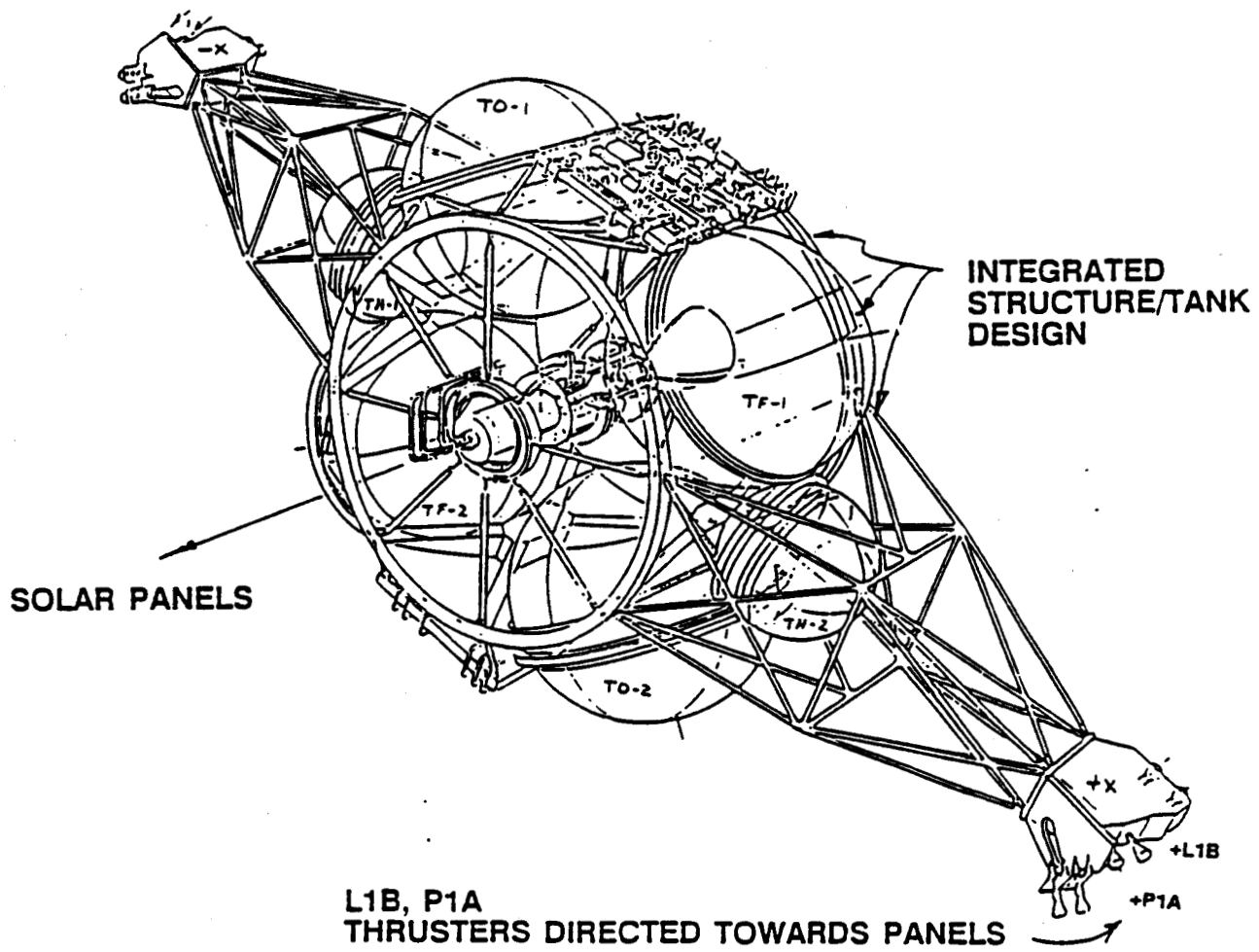sequence assessments; they could seriously impact spacecraft functions and
performance.

INTEGRATED
STRUCTURE/TANK
DESIGN

SOLAR PANELS

L1B, P1A
THRUSTERS DIRECTED TOWARDS PANELS

Figure 3.   Galileo Retro Propulsion Module

Galileo
SCIENCE ISSUES

SHOW STOPPERS

- GALILEO SCIENCE RETURN REQUIRES
  - PROBE
  - ORBITER MISSION

- IF CANNOT DELIVER PROBE AND ACHIEVE ORBIT - MISSION VALUE IS TOO LOW TO PROCEED

CONCERNS/SERIOUS ISSUES

- SPINNING SCIENCE
  - FOV LIMITATIONS
  - S/C CHARGING
  - EMI

- DESPUN SCIENCE (SSI, NIMS, PPR, UVS)
  - POINTING DEGRADATION
  - FOV LIMITATIONS
    - CONSTRAINTS ON HI SOLAR PHASE OBSERVATIONS

  - NO SCIENCE TURNS - ELIMINATES / LIMITS:
    - HI RES. RING SCIENCE
    - Io MONITORING
    - DARKSIDE SATELLITE OBSERVATIONS
    - SOLAR OCCULTATION OBSERVATIONS
      - AURORA
      - LIGHTNING

CHAPTER II

F. SCIENCE

SECTION F

GALILEO SCIENCE ISSUES

1.   SCIENCE OBJECTIVES AND FLYBY/PROBE CONSIDERATIONS

The study considered changing the Galileo spacecraft to a flyby/probe configuration to reduce the mass penalties associated with a solar retrofit. However, the study concluded that a change from the orbiter/probe configuration to a flyby/probe configuration would entail a sacrifice of scientific information significant enough to reduce the mission value to a level not worth doing. The Pioneer and Voyager missions have already derived much of the photographic, spectroscopic, magnetic, and gravitational information pertaining to Jupiter that a flyby mission is capable of deriving. With the orbiter, Galileo is supposed to derive much more comprehensive information regarding the nature of Jupiter's atmosphere, its energy emissions in excess of what it absorbs, its magnetosphere and charged particle interactions, its gravitational field, and Io's volcanic activity. Without the orbiter, this more comprehensive information could not be acquired; a flyby would be essentially a replay of the Pioneer and Voyager missions.

2.   SOLAR RETROFIT ISSUES

Even retaining the orbiter/probe configuration, however, a solar retrofit would degrade Galileo's science capabilities. This degradation would result from two causes: the solar array surface area and the array mass.

At 210 square meters, the solar array surface area would present serious field-of-view (FOV) limitations and static charging concerns. The FOV limitations would affect the plasma, energetic particle, dust, and solar phase observations. If static charging of the arrays occurred, the plasma science observations would be degraded.

The array mass would present itself as a science constraint by increasing the spacecraft's inertia to the point where spacecraft science turns would not be possible for the amount of available propellant. Thus, a solar retrofitted Galileo would not be able to conduct high resolution ring science, Io monitoring, darkside satellite observations, or Jupiter aurora and lightning observations.

CHAPTER II

G. SYSTEM ISSUES SUMMARY

SECTION G

SYSTEM ISSUES SUMMARY

1.   CONFIGURATION AND MASS CHANGES

Configuration changes and large mass increases are the direct drivers to system design issues. They create a set of severe system design problems. This can be anticipated by observing that the large solar panels dominate the configuration.

2.   STRUCTURE REDESIGN

Because of the very large arrays and the volume of batteries that are required, structure redesign is significant. Bus electronics volume increases would be a minimum of 30%. The configuration would leave science fields of view constrained, including likely obscuration of the relay radio antenna view of the probe during its descent into the Jupiter atmosphere.

3.   FAULT PROTECTION AND CONTROL SOFTWARE

Although new designs would be required for much hardware, a very severe system issue that must be dealt with is the system fault protection, complicated by the changing thermal environment and power levels which do not result in a fail safe mode.

If currently available solar arrays are used, the spinning inertia would be increased by a factor of 18. This drives propellant requirements, and would create an entirely new and difficult control analysis and design problem. Most of the control software would need to be rewritten. Pointing performance and stability would likely be degraded, affecting science value, costs, and maneuver propellant allocations. New modes for the data system, although straightforward, would need to be developed.

4.   PROPELLANT CAPACITY

Because increasing the propellant tank capacity introduced more mass problems, the tank size was kept unchanged. Plume impingement and contamination problems would result from the configuration.

5.   TEMPERATURE CONTROL

Temperature control would be very difficult, especially when trying to work the fault protection, since too much or too little sun on the solar panels could be mission catastrophic, and the limits would change as a function of mission phase.

Table III. GLL Solar Power Retrofit Dry Mass Deltas

Subsystem	Item	Mass(KG)		
		Rigid	SAFE	APSA
STRUCTURE				
	Remove RTG booms, outriggers, transition truss, mounting bracket and deployment mech.	-15		
	Core structure beefup for S/C, Propulsion module, and Adapter	60	40	20
	Panel booms, outriggers, and deployment mechanisms	75	35	20
	Doghouse structure for batteries	15 (52)*		
POWER				
	Remove RTGs	-112		
	Solar panels (cells and substrate for 30kW earth 640W Jupiter, 210 sq. m.)	650	420	210
	Batteries (4 hrs. @ 450W NiCd)	140 (490)*		
	Electronics delta	25		
ATTITUDE CONTROL				
	Actuators to rotate and articulate three panels (delta of 4 @ 3 kg)	12		
TOTAL		850 (1237)	561 (948)	316 (703)*

*Candidate GLL missions require up to 14 hours for Jupiter occultation, which would require batteries sized at 3.5 times greater than the 4 hour assumption.

6. ON-BOARD HEAT SOURCES

The GLL Probe would be separated from the Orbiter five months before entry. It carries 36 Radioisotope Heating Units (RHUs). There is no other possible heat source for the Probe, so the RHUs were assumed kept in the design for the Orbiter and the Probe. The total RHU complement is 120.

7. RETEST REQUIREMENTS

Full retesting as well as complete rework of the mission, sequences and mission operations would be needed.

8. BATTERY MASS INCREASE

Mass increases would be very large. A four hour maximum occultation was assumed to size the batteries. Candidate GLL missions have been observed with occultations as high as 14 hours which would demand even more unreasonable mass growth for batteries. Some mission trades would have to be made to properly size the batteries.

9. CONCLUSIONS

There is no "easy GLL retrofit" even with very significantly degraded mission requirements. Trying to fly a reworked GLL in a few years simply CANNOT BE DONE.

CHAPTER III

EXISTING TECHNOLOGY BASE

AND

PROSPECTS FOR THE FUTURE

Power Source Technology Status

Agenda

Introduction

Nonisotopic

Photovoltaic
Advanced Solar Dynamic (ASD)

Isotopic

RTGs

Alkali Metal Thermoelectric Generator (AMTEC)
Turbine Electric Conversion (TEC)

Summary/Recommendation

1.   AGENDA

    a.   <u>Introduction.</u> This presentation will discuss the various types of nonisotopic and isotopic sources available for use on a planetary mission in the 1990's. It is important to note that there are only two types of sources available for future missions, those that use the solar insolation and those that use radioisotopes for the power system [VanLandingham, 1988, Ref. 7].

    b.   <u>Nonisotopic.</u> Two types of nonisotopic systems will be discussed, photovoltaics and advanced solar dynamic (ASD). In the photovoltaic system, energy from the sun is converted directly to electricity. This type of source has been available for a number of years and has a proven flight history. Advanced solar dynamic systems utilize the energy from the sun to heat a working fluid which in turn is converted to electricity utilizing rotating machinery.

    c.   <u>Isotopic.</u> Isotopic sources utilize the decay heat process of a radioisotope to power a conversion process. In the Radioisotope Thermoelectric Generator, the heat is converted directly to electricity utilizing solid state materials. The AMTEC system utilizes a working fluid also in a direct energy conversion technique. The TEC system is similar to the ASD but the power source is a radioisotope rather than the sun.

    d.   <u>Summary/Recommendation.</u> The discussion will conclude with a summary of the technology and its flight readiness date.

Introduction

Objective

To investigate the technology readiness of advanced power sources for planetary missions

To recommend earliest launch readiness dates for the various advanced power source technologies

2. OBJECTIVE

The objective of this presentation is to investigate the technology readiness of advanced power sources for planetary missions. This will involve a detailed review of the various sources available for planetary missions, their current status, and projected flight readiness dates if available. Particular attention will be paid to readiness for a mid to late 1990 launch to Jupiter although the application to the remaining far outer planets will also be discussed.

The presentation will also discuss when the technologists feel that the advanced power sources will be available for flight. A key element to this decision will be whether any flight demonstrations have been conducted for new technologies, or whether the new source is a modification of those already qualified and flown.

CHAPTER III

A.   NONISOTOPIC TECHNOLOGIES

Photovoltaic Cell Status

Silicon

Well established with long history of use. Currently achieving about 14% efficiency with large production capability

GaAs

Limited manufacturing capability. To date very limited flight experience.

Next generation cells on Germanium. Even more limited production capability and no flight experience. These cells will be space ready in the early to mid 1990's.

InP

Next generation of super cell. Currently in the laboratory phase. Space readiness in late 1990's.

Conclusion - *Silicon is only near term option. GaAs holds promise for future but its true benefits at Jupiter not well characterized yet.*

3.   PHOTOVOLTAIC CELL STATUS

a.   Introduction. Before there is a discussion of the nonisotopic sources and particularly the photovoltaic systems, a detailed discussion of the cell status in order. The current and projected status of Silicon, GaAs, and InP (Indium Phosphide) will be discussed.

b.   Silicon. There is a long history of success with Silicon solar cells. Their degradation and operation over wide-ranging conditions are well understood. While they do not have the high efficiency projections of GaAs or InP, they are well respected as the workhorse cell for the near future.

c.   GaAs. This is still a young cell in terms of its production capability and flight experience. However, it clearly is the cell for high efficiency and radiation tolerance in the early 1990's. Current projections are that efficiencies of up to 20% are achievable.

d.   InP. Indium Phosphide is the next generation of high efficiency cell. This combination of solid state materials holds promise for much higher efficiencies. Unfortunately, this technology will not be ready until late into the 1990's.

Photovoltaic Power Sources

Planar

Current SOA - Rigid Arrays flown on Magellan, Mars Observer, TOPEX. Power density very low 35 W/kg

SAFE- The Solar Array Flight Experiment demonstrated design of 70 W/kg. Has not been flown on a civilian mission, no experience.

APSA- The Advanced Photovoltaic Solar Array is currently in fabrication. Due to complete ground demonstration in 1991.

Concentrator

Current SOA - Rigid systems built and tested on earth. Specific power less than current rigid systems.

Advanced - Flexible, light weight, high performance space systems are only in the conceptual design phase. No detailed or demonstration of concentrator designs planned.

Conclusion - Concentrator arrays are not a proven technology. Planar advanced arrays are the only solution for spacecraft power to Jupiter. Concentrator arrays may be used beyond Jupiter but large development costs are needed.

4.   PHOTOVOLTAIC POWER SOURCES

There are two types of arrays suitable for use, planar arrays and concentrator arrays.

a.   Planar. The current state of rigid arrays that have flown on Magellan, and which will fly on Mars Observer and TOPEX have a specific power density of only 35 W/kg. For a 30 kW array this is prohibitive.

The Solar Array Flight Experiment (SAFE) array is a much lighter array, about 70 W/kg. It has flown on the shuttle as an experiment, but there is no experience on civilian flights. Thus, a reasonable amount of design time would be required to adapt the flight experiment into an array suitable for the mission to Jupiter, but the design process would be straightforward.

The Advanced Photovoltaic Solar Array (APSA) is a high performance, lightweight array design. Ground demonstrations thus far have shown a specific power of 130 W/kg. This is clearly the array of choice for missions to Jupiter. Unfortunately, it is not scheduled for completion of the ground demonstration until 1991 and a flight experiment in 1993. Thus, flight readiness will not be until the mid 1990's [Kurland/Stella, 1988, Ref. 8].

b.   Concentrator. Current rigid concentrator systems are very bulky and based upon plastic designs. In the radiation environment of space, the plastic surfaces will become opaque and the concentrator system will fail.

The advanced designs are proposed as flexible lightweight systems [Rockey, 1981, Ref. 9]. Unfortunately, these systems are only conceptual designs. There have been no detailed designs or ground demonstrations of such a system, nor are any planned for the near term.

Advanced Solar Dynamic Power Sources

NASA Goal

20 W/kg for Brayton and Stirling Systems

Technology Status

Ground Demonstrations of space Solar Dynamic Power Sources will not be until mid to late 1990's

Spacecraft Issues

The fast rotations of the alternators (>20,000 rpm) imply new attitude control issues for stability and turning the spacecraft.

Conclusion - *Solar Dynamic Systems do not have near term application for planetary spacecraft to Jupiter or beyond due to technology readiness for a late 1990's launch.*

5.   ADVANCE SOLAR DYNAMIC POWER SOURCES

For the Advanced Solar Dynamic Power Sources, the NASA goal is for 20 W/kg. For a 1 kw source at Jupiter, this source would weigh 200 kg. However, the ASD sources are not yet close to technology readiness. The earliest ground demonstrations will not be until the mid to late 1990's. Thus, there will not be any ASD systems ready until the first decade of the next century for a flight to Jupiter [NASA, 1988, Ref. 10].

Since the ASD system involves a rotating machine, there will be new integration and spacecraft attitude control issues. The inertial mass of the ASD can imply difficulty in turns and in slewing maneuvers for the spacecraft. At this time, all of these can be overcome with careful attention to design of the spacecraft and its operators.

Thus, while ASD could be an application for a planetary mission to Jupiter, it seems unlikely until the 21st century.

CHAPTER III

B. ISOTOPIC TECHNOLOGIES

Radioisotope Thermoelectric Generators

Current Status

> Current Galileo RTG efficiencies are 6.5% beginning of mission.
> Specific power is 5.3 W/kg

Advanced RTGs

> Modular RTGs will utilize new technology to achieve specific power of
> 7.7 W/kg. The modular design will allow closer match between source
> and load requirements

Advanced Materials

> 200% increase in material performance projected for completion in mid
> 1990's. This would allow a 200% decrease in thermal requirements from
> a radioisotope source.

Conclusion - *RTGs are the only power source suitable for outer planet missions for at least the next decade. Improvements will be made to reduce the amount of isotopic material required.*

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6. RADIOISOTOPE THERMOELECTRIC GENERATORS

Isotopic power sources utilize the decay heat from a radioisotope to produce electricity. This decay heat is either converted directly to electricity or indirectly through a rotating machine. The first discussion will be about a Radioisotope Thermoelectric Generator (RTG). It utilizes a solid state converter to convert the heat directly to a low voltage power. By placing enough of these converters in series and parallel, power at 30 Volts dc can be achieved.

a. <u>Current Status.</u> RTGs like those currently on Galileo and Ulysses operate at about 6.8% efficiency. This gives a specific power of about 5.3 W/kg or 110 kg. Thus, if the decision is made strictly on a weight basis, RTGs are a very lightweight system.

b. <u>Advanced RTGs.</u> The Modular RTG concept (MOD-RTG) will utilize technology to achieve a specific power of 7.7 W/kg. The modular design will allow a closer match between spacecraft needs and the power source. In this way, the amount of Pu 238 can be reduced [Hartman, 1988, Ref. 11].

c. <u>Advanced Materials.</u> In the mid 1990's new materials will allow for a 200% increase in the efficiency and thus the specific power. In this way, the RTG will still be competitive with the ASD and APSA systems of the mid 1990's. The difference will be that this design will be qualified by similarity, meaning that devices similar in nature have flow and have a proven flight record. The APSA and ASD system, on the other hand, will need to be qualified from the piecepart up, a very costly and time-consuming activity [NASA, 1988, Ref. 10].

Thus, the RTG system will continue to be the backbone power source of the far outer planet mission set. It has a proven record of reliability and safety. The improvements that are planned will allow for less Pu-238 to be flown, thus reducing its cost even further.

Alkali Metal Thermoelectric Converter (AMTEC)

Current Status

Laboratory model developed and electrodes undergoing life tests. Initial results show 20% cell efficiency. Translates to a specific power of 20 W/kg.

Future Plans

Test cells and modules developed in early 1990's.

Life verification of electrodes (50,000 hours) completed in mid 1990's.

Module life of 10,000 hours verified by mid 1990's.

Conclusion - AMTEC is not ready yet for flight implementation. By the late 1990's it will be a viable contender for outer planet missions.

7.   ALKALI METAL THERMOELECTRIC CONVERTER (AMTEC)

The Alkali Metal Thermoelectric Converter, or AMTEC, is also a direct energy conversion system. It utilizes the heat from a isotope source to produce a flow of electrons through sodium. Although the heat source could be something other than a radioisotope, this seems to be the most compact design. A system using solar heat would need a concentrator and heat pipes to conduct the heat to the AMTEC cells.

a.   Current Status. A laboratory cell has been developed and is in the process of undergoing electrode life tests. Initial results have shown a cell efficiency of 20% which translates to a specific power of about 20 W/kg. Notice that this is higher than the MOD-RTG and advanced materials RTGs. Thus, this device seems to be a good candidate power source for far outer planet missions [Bankston, 1989, Ref. 12].

b.   Future Plans. Engineering model Test Modules will be developed in the early 1990's. These will be used for life verification of the electrodes up to 50,000 hours and Test Modules up to 10,000 hours in mid 1990's. In addition, models will be developed to predict life for future use in flight project power source projections. These models will be verified against the actual data obtained from the cell and Test Module tests. A flight experiment in the mid 1990's will be needed to verify zero G operation of the fluid loop.

AMTEC is a viable source for missions in the late 1990's. It will have shown technology readiness by the mid 1990's with a reasonable influx of dollars. To get to Level 6 readiness will take $30M while getting to Level 8 will take $100M.

Turbine Energy Conversion

Current Status

Initial conversion studies and preliminary design completed.

Projected specific power is 5.2 W/kg.

Smallest projected size is about 1 kW_e.

Future Plans

Component tests completed in early 1990's.

Engineering Unit life tests will be completed in the mid 1990's.

CONCLUSION - Turbine energy will not be available for outer planet missions until the late 1990's. Its benefits over AMTEC will be evaluated at that time.

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8.   TURBINE ENERGY CONVERSION

It is also possible to use a radioisotope heat source to power a turbine system. In this case, the low temperature heat source would replace the high heat source of the advanced solar dynamic systems. Because of this large difference in heat sources, a total redesign and qualification process would be required.

a.   Current Status. The initial preliminary and laboratory models have been completed. The two designs have focused on a Closed Brayton Cycle and the Rankine Cycle systems. Tests results obtained for the Rankine Cycle have achieved 1000 hours. While this is far short of the 67,000 hours requirement, it does show that the concept seems to be viable. Projected specific power is about 50 W/kg for a 1 kW unit. This would be somewhat too big for a planetary mission, and scaling would be an issue [Bennett, Gary L., 1988, Ref. 13].

b.   Future Plans. The individual component life test will be completed in the very early 1990's. Schedules indicate that the Rankine Cycle life tests will be completed in 1992-1993. For the Closed Brayton Cycle, the engineering units will not complete life tests until the mid 1990's.

Thus, the turbine-energy systems do not seem to be well positioned for consideration until the late 1990's. Even so, the advantages of this system over AMTEC are not clear at this time. A more detailed evaluation will need to be made in the mid 1990's to evaluate the technology readiness of each and its applicability for future flight missions.

CHAPTER III

C. SUMMARY FINDINGS

Summary

Solar Array advances will not allow far outer planet missions, RTGs will be the only choice for the short term

Solar Array advances could provide useful technology for mid 1990's launch to Jupiter

RTG material advances will also provide useful technology for mid 1990's launch

AMTEC will not be ready until the late 1990's for flight programs

Dynamic and turbine energy systems will not be ready until late 1990's

9. SUMMARY

It is clear, that for missions beyond Jupiter, the only power source for a 1990's launch will be a Radioisotope Thermoelectric Generator (RTG). The solar insolation is so weak at that distance from the sun that direct conversion using solar power becomes untenable. In addition, as improvements are made in the RTGs, from MOD-RTGs to advanced materials, the amount of Pu 238 will continue to decrease.

The Advanced Photovoltaic Solar Array appears to be a viable mid 1990's or later option for flights up to and including Jupiter. While the arrays would be very big, the additional power at the Earth would permit the consideration of electric propulsion techniques. This would decrease the trip times to Jupiter, Mars, or solar polar orbits, but introduce a new set of spacecraft design issues.

The hope for the future of far outer planet missions will be AMTEC, turbine-energy and dynamic systems. Because of its direct energy conversion techniques, AMTEC is the front runner today. However, because these systems are now only in the preliminary design phases, with engineering unit testing under way, it is difficult to predict a clear choice at this time. It is anticipated that each flight project between now and the year 2000 will investigate the use of these sources against conventional RTGs for far outer planet missions.

In conclusion, there are no power sources available to a project manager for missions to Jupiter and beyond except RTGs. This will clearly be the source of choice for planetary spacecraft through the mid 1990's. The Mariner Mark II class of spacecraft has chosen for its first two missions in 1995 and 1996 an RTG-based system. While AMTEC and solar were investigated for these launches, the high technology that would meet the missions' requirements was unavailable. Thus, RTGs became the choice due to their favorable history, reliability, and mass.

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CHAPTER IV

SPACECRAFT DESIGN PROCESS

AND

IMPLICATIONS

CHAPTER IV

SPACECRAFT DESIGN PROCESS AND IMPLICATIONS

1.   OVERVIEW

It has been shown that there is no "feasible retrofit" of GLL that could be launched in the 1989 to 1991 timeframe. Furthermore, it is clear that a retrofit of GLL to make it a solar spacecraft would be more difficult than developing and launching Mariner Mark II (MMII), starting today. It is thus concluded that there is no way to do that task even in a few years beyond 1991, and if it were attempted, it would be a very expensive and technically risky undertaking.

Considering the problems that have been identified with a solar retrofit approach, this study has concluded that the only appropriate way to proceed with a solar mission to Jupiter is to consider the inheritance that is available from Mariner Mark II, GLL, and other projects, and start a design process that minimizes the anticipated problems. In other words, use what is possible from GLL (which probably would not be much) and design and develop a new solar spacecraft for a Jupiter mission. It would almost surely be three-axis stabilized and would employ some new technology to reduce cost and risk.

2.   OTHER FACTORS

a.   The project would need a full pre-project effort.

b.   A significant technology development would probably pay large dividends.

c.   A standard pre-project development would help assure maximum mission return for the investment at acceptable risk.

d.   Funding would force changes in other currently planned projects.

e.   Launch would not be until at least 1996, and probably 1999 or later.

CHAPTER IV

A. OVERVIEW

OVERVIEW

o   **MMII WELL DEFINED**

   -   **TIGHT SCHEDULE TO LAUNCH IN '95**

o   **GLL VERY TIGHT SCHEDULE TO DEFINE AND MAKE "MINOR" MODIFICATIONS IN 3 1/2 YEARS FOR '89 LAUNCH**

o   **GLL SOLAR DESIGN WOULD BE MAJOR NEW DESIGN TASK**

   -   **MORE APPROPRIATE TO REDESIGN CONSIDERING INHERITANCE & TECHNOLOGY OPTIONS THAN TO REWORK EXISTING GLL SPACECRAFT**

CHAPTER IV

B. PROGRAM DEVELOPMENT PROCESS

SOLAR POWERED JUPITER ORBITER PROBE

PROGRAM DEVELOPMENT PROCESS

o PRE-PROJECT DEVELOPMENT EFFORT REQUIRED

- PROGRAMMATIC

    - PROJECT SCIENCE GROUP REQUIRED TO REVIEW
      SCIENCE OBJECTIVES AND PROVIDE INPUT ON
      MISSION AND SPACECRAFT DESIGN, SCIENCE
      PAYLOAD MODIFICATION

    - MISSION DESIGN TEAM TO REVIEW/REVISE
      MISSION DESIGN

    - SPACECRAFT DESIGN TEAM TO SYNTHESIZE
      PRELIMINARY SPACECRAFT DESIGN

<u>SOLAR POWERED JUPITER ORBITER PROBE</u>
<u>PROGRAM DEVELOPMENT PROCESS</u>

o   PRE-PROJECT DEVELOPMENT   (CONTINUED)

-   TECHNOLOGY

    -   DEVELOPMENT PROGRAM REQUIRED

        -   LARGE DEPLOYABLE SOLAR ARRAYS

        -   POWER PROCESSING EQUIPMENT

        -   ELECTRIC PROPULSION

        -   LIGHTWEIGHT BATTERIES

SOLAR POWERED JUPITER ORBITER PROBE
PROGRAM DEVELOPMENT PROCESS

o **PRE-PROJECT DEVELOPMENT SCHEDULE AIMED TO A NON-ADVOCATE REVIEW**

- **SCHEDULE DEPENDS ON TECHNOLOGY DEVELOPMENT (PROBABLY 3 - 5 YEARS)**

o **NORMAL 4 - 5 YEAR SPACECRAFT DESIGN, DEVELOPMENT, INTEGRATION AND TEST CYCLE**

- **INHERITANCE FROM MMII AND GLL DIFFICULT**

- **PROPULSION MODULE FOR LARGE DELTA V**

- **PARTS OBSOLESCENCE (FROM GLL) AND RADIATION SENSITIVITY (FROM MMII) ISSUES (LACK OF QUALIFIED MICROPROCESSOR)**

SOLAR POWERED JUPITER ORBITER PROBE
PROGRAM DEVELOPMENT PROCESS

o   **FUNDING**

- **SIGNIFICANT PRE-PROJECT INVESTMENT REQUIRED ($50M TO $100M)**

  - **CODE R PARTICIPATION REQUIRED**

  - **PARTIAL FUNDING BY GLL MOS ALLOCATIONS**

- **MMII CLASS EFFORT ($1B)**

- **CURRENT FUNDING WEDGE WILL NOT ACCOMMODATE**

  - **REPRIORITIZATION OF EXISTING PROGRAMS REQUIRED**

CHAPTER IV

C. SCHEDULE

SCHEDULE

89 90 91 92 93 94 95 96 97 98 99

PRE-PROJECT

CONGRESSIONAL APPROVAL

DEVELOPMENT

LAUNCH
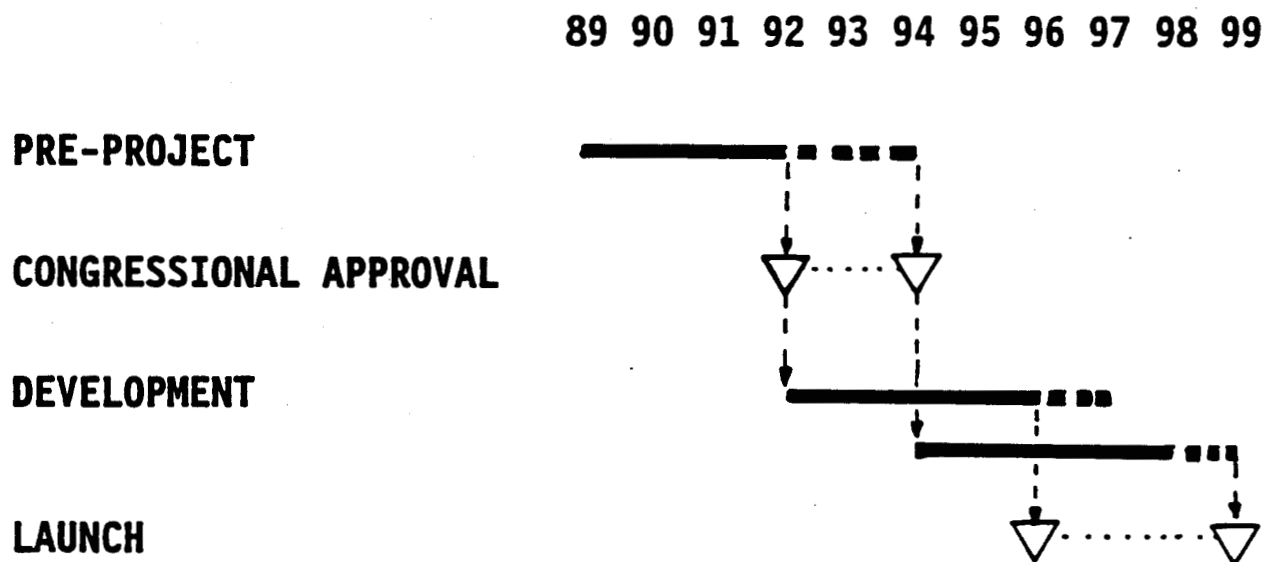
CHAPTER V
CONCLUSIONS

V. STUDY CONCLUSIONS

1.   RETROFITTING GALILEO WITH SOLAR ARRAYS

This study has determined that two basic problems, mass and schedule, prevent the retrofit of Galileo with solar arrays from being a viable mission option. Each of these two problems comprises a variety of component difficulties -- these component difficulties involve spacecraft configuration and structure, spacecraft power, attitude control and dynamics, the propulsion module, and mission science capabilities.

a.   <u>Configuration and Structure Issues.</u> Retrofitting Galileo with solar arrays would necessitate making a variety of structural changes to the spacecraft. Each of these structural changes would add to the total launch weight. For instance, the core structure for the spacecraft, propulsion module, and adaptor would have to be strengthened to accomodate the additional load imposed by the solar arrays -- a change that would add 60 Kg of mass for the rigid arrays and 20 Kg for the Advanced Photovoltaic Solar Arrays (APSA) arrays. In addition, the panel booms, outriggers, and deployment mechanisms necessary for the solar arrays would contribute 75 Kg for the rigid arrays and 20 Kg for the APSA arrays. These masses contribute to a total launch weight that exceeds Shuttle/Inertial Upper Stage (IUS) launch capabilities for the current Galileo orbiter/probe configuration.

Many of these changes in spacecraft configuration and structure also require more time to implement than is available for a 1989 or 1991 launch. Some of these changes include modification of the spacecraft's core structure, modifying the IUS to handle the new mechanical demands and stresses imposed by the reconfigured spacecraft, design of a new power-shunt radiator system, and the design of solar panel deployment mechanisms.

b.   <u>Power Issues.</u> The power system additions associated with attaching solar arrays to the Galileo spacecraft would contribute significantly to the total launch weight. Rigid solar arrays, alone, would add 650 Kg to the spacecraft. APSAs would add 210 Kg. Because of occultation with Jupiter, at least 140 Kg of batteries would be needed for the dark-side operations of an orbiter/probe configuration. And, an additional 25 Kg of electronics would be needed to support array and battery operations. These masses constitute a large portion of the total launch weight that would be associated with the retrofitted spacecraft -- weight in excess of current Shuttle/IUS launch capabilities.

Even if mass were not an issue, the power subsystem would have two significant schedule problems. First, neither the rigid solar arrays nor the APSA arrays would be available for a 1989 or 1991 launch. Rigid arrays suitable for the Galileo mission would require at least one year to design and two years to manufacture. The APSA arrays would require at least one year to design and three years to manufacture. Additional time would be required to test the arrays and incorporate them into the spacecraft. The second problem would be the need to redesign the power subsystem and all of the fault protection. These redesign tasks would delay Galileo's launch to a date beyond the 1989 and 1991 launch windows.

c.   Attitude Control and Dynamics Issues. Because of the solar panel masses and sizes required for a mission to Jupiter, the inertia and angular momentum associated with Galileo's dual-spin design would increase substantially. This increase would slow precession and spin-up maneuvers and would vastly increase propellant consumption. The increase in propellant consumption would then necessitate carrying more propellant on board Galileo -- a necessity that would translate into a propellant mass increase of 522 Kg for rigid arrays and 122 Kg for APSA arrays. This increased propellant mass represents yet another portion of a total launch mass that would exceed current Shuttle/IUS launch capabilities for the orbiter/probe configuration.

Additional problems inherent to a solar retrofitted Galileo would be the control of solar panel power and temperature levels. New actuators and sun sensors would have to be designed and installed to ensure that the angle of the solar panels could be adjusted to keep power and heat levels constant throughout the Venus-Earth-Earth-Gravity-Assist (VEEGA) trajectory -- a trajectory that carries the spacecraft first near to and then far from the sun. In addition, new control algorithms, software, and fault protection would have to be developed to support operation of the actuators and sensors. These developments probably could not be completed in time for a 1989 or 1991 launch.

d.   Propulsion Module Issues. While the study could not identify any retrofit-related propulsion module problems that would prevent a solar mission, it did identify two serious concerns requiring further consideration: plume impingement and extended maneuver sequences. To the extent that the exhaust plumes from the spacecraft's thrusters could impinge on the solar panels, the potential for array degradation could exist. Hence, additional array might be required to offset this degradation -- array that would simply add to all of the mass and schedule problems discussed previously. As for the extended maneuver sequences, the current Federal Republic of Germany-supplied 10 newton (N) thrusters now have limited firing sequences so as to remain within specified thermal bounds. However, a solar retrofitted Galileo would be characterized by high mass/inertias that would necessitate firing sequences which would exceed the current 10-N thruster thermal constraints. Thruster redesign by the Federal Republic of Germany to remove the 10 N constraints could pose serious schedule delays.

e.   Science Issues. The study considered changing the Galileo spacecraft to a flyby/probe configuration to reduce the mass penalties associated with a solar retrofit. However, the study concluded that a change from the orbiter/probe configuration to a flyby/probe configuration would entail a sacrifice of scientific information significant enough to reduce the mission value to a level not worth doing. The Pioneer and Voyager missions have already derived much of the photographic, spectroscopic, magnetic, and gravitational information pertaining to Jupiter that a flyby mission is capable of deriving. With the orbiter, Galileo would derive far more comprehensive information regarding the nature of Jupiter's atmosphere, its energy emissions in excess of what it absorbs, its magnetosphere and charged particle interactions, its gravitational field, and Io's volcanic activity. Without the orbiter, this more comprehensive information could not be acquired; a flyby would be essentially a replay of the Pioneer and Voyager missions.

Even retaining the orbiter/probe configuration, however, a solar retrofit would degrade Galileo's science capabilities. This degradation would result from two causes: the solar array surface area and the array mass. At 210 square meters, the solar array surface area would present serious field-of-view (FOV) limitations and static charging concerns. The FOV limitations would affect the plasma, energetic particle, dust, and solar phase observations. If static charging of the arrays occurred, the plasma science observations would be degraded.

The array mass presents itself as a science constraint by increasing the spacecraft's inertia to the point where spacecraft science turns are not possible for the amount of available propellant. Thus, a solar retrofitted Galileo would not be able to conduct high resolution ring science, Io monitoring, darkside satellite observations, or Jupiter aurora and lightning observations.

        f.   **Issue Implications.** The issues associated with spacecraft configuration and structure, power, attitude control and dynamics, propulsion, and science lead to the following conclusions:

      1)   If retrofitted with either rigid or APSA solar arrays, the Galileo spacecraft, as an orbiter/probe combination, would weigh too much to launch into the proper trajectory with the Shuttle/IUS combination.

      2)   A flyby/probe mission would still involve too much mass with rigid solar arrays, and, in any event, would not satisfy enough of the mission's objectives to be worth the cost. In fact, even a retrofitted Galileo orbiter/probe, were one possible, would not be capable of fulfilling all the scientific objectives that the current, RTG-powered mission is capable of fulfilling.

      3)   Neither a rigid solar array retrofit nor an APSA array retrofit could be completed in enough time to allow a 1989 or 1991 launch.

In view of the insurmountable mass and schedule difficulties associated with a solar retrofit of Galileo, the study team concluded that the only alternative to an RTG-powered Galileo mission would be to cancel the Galileo mission and design a completely new, solar-powered spacecraft for the late 1990s.

    2.   EXISTING POWER TECHNOLOGIES AND THEIR IMPLICATIONS FOR THE FUTURE

        To the extent that cancellation of the Galileo mission and initiation of a completely new, solar-powered mission to Jupiter might be considered, the study examined the current status of power technologies to see which technologies might best support such a mission. This examination revealed that, among the many solar technologies, Advanced Photovoltaic Solar Arrays (APSAs) show the most promise for a late 1990's Jupiter mission. However, no solar technology demonstrated any viability for missions more distant than Jupiter. As for batteries and fuel cells, no technology exists that will deliver enough energy per pound of mass to make it suitable as a sole power source for any planetary mission. However, a new type of radioisotope thermoelectric converter known as

an Alkali Metal Thermoelectric Converter (AMTEC) is being developed that may require far less plutonium-238 fuel than an RTG using thermocouples -- thereby reducing the risk as well as the cost. By the late 1990's, AMTECs might be capable of serving as power sources for outer planet missions.

3. THE SPACECRAFT DESIGN PROCESS AND ITS NEW MISSION IMPLICATIONS

The spacecraft design process for a new, solar-powered mission to Jupiter would have to proceed in two stages. The first stage, pre-project development, would involve science objective development, mission design, spacecraft design, and technology development. These pre-project activities would take three to five years. The second stage, normal project development, would involve further spacecraft design, development, integration, and testing. These activities would require an additional four to five years.

For a completely new, solar-powered mission to Jupiter, this process and its associated time requirements indicate that at least seven to ten years of pre-launch activity would be required. Hence, a new Jupiter mission could not be launched earlier than 1996 and probably could not be launched until 1999 or later. On the basis of past project experience, the study estimated that pre-project development would probably cost $50 million to $100 million. Normal project development would probably cost at least another $1 billion.

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